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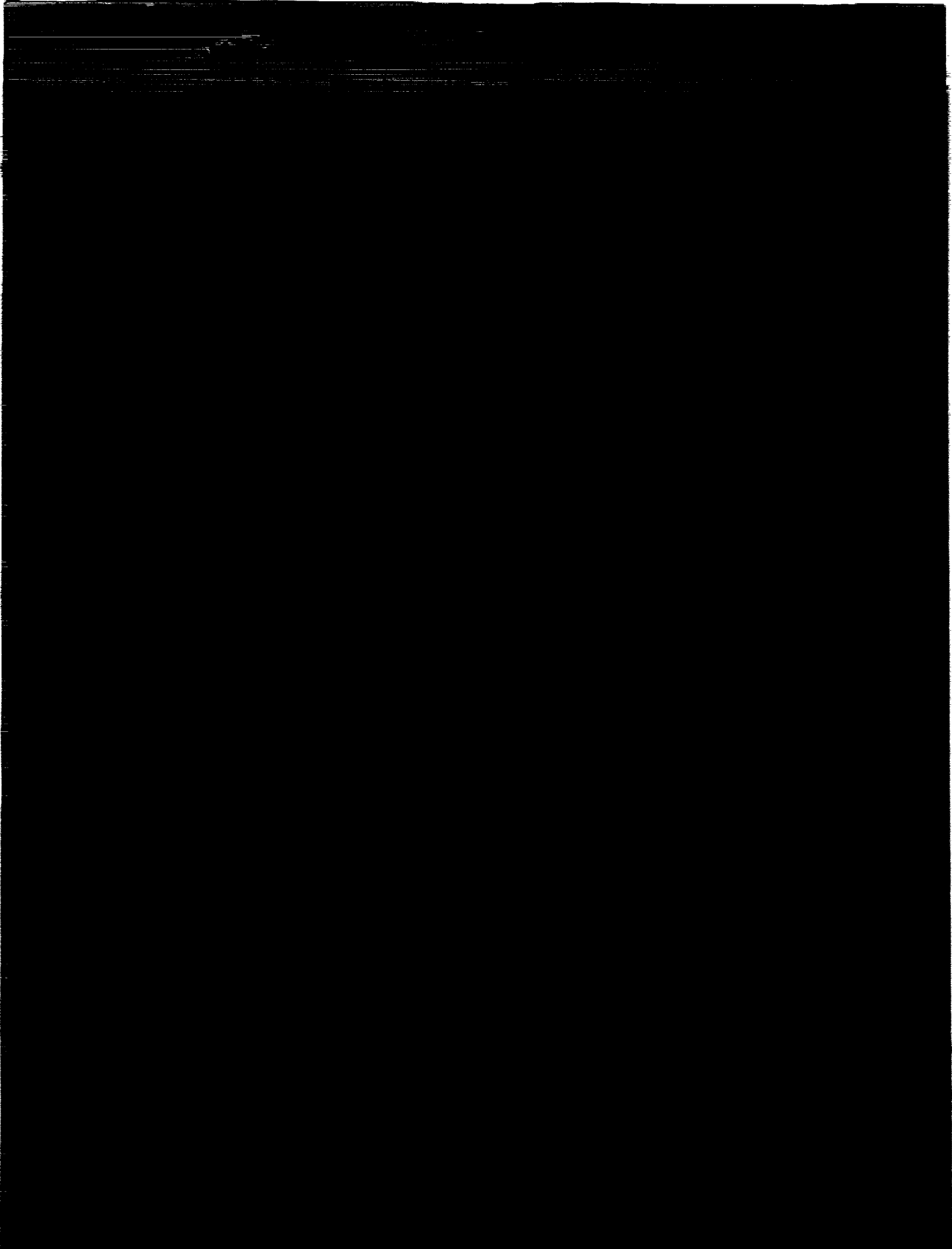
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Enhancing Space Transportation: The NASA Program To Develop Electric Propulsion

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ABSTRACT

The NASA Office of Aeronautics, Exploration and Technology (OAET) supports a research and technology (R&T) program in electric propulsion to provide the basis for increased performance and life of electric thruster systems which can have a major impact on space system performance, including orbital transfer, stationkeeping, and planetary exploration. The program is oriented toward providing high-performance options that will be applicable to a broad range of near-term and far-term missions and vehicles. The program, which is being conducted through the Jet Propulsion Laboratory (JPL) and Lewis Research Center (LeRC), includes research on resistojets, arcjets, ion engines, magnetoplasmadynamic (MPD) thrusters, and electrodeless thrusters. Planning is also under way for nuclear electric propulsion (NEP) as part of the Space Exploration Initiative (SEI).

INTRODUCTION

Onboard propulsion is essential for both commercial and government spacecraft. Onboard propulsion uses include orbit circularization (apogee motors), north-south stationkeeping (NSSK) for geosynchronous Earth orbit (GEO) spacecraft, orbit control, and trajectory modifications (Byers 1989). From Figure 1 it can be seen that propellant comprises about 76% of the mass of a planetary spacecraft such as the planned Mariner Mark II Comet Rendezvous Asteroid Flyby (CRAF) spacecraft. For the Galileo spacecraft the propellant mass fraction is 43% (Palaszewski and Englebrecht 1987). Future commercial and government spacecraft are expected to be even more demanding for propulsion and power.

Clearly there is a need to develop more fuel-efficient (i.e., high specific impulse) onboard propulsion systems to lighten spacecraft and/or to provide more room for additional payload and/or to reduce trip time. Since onboard propulsion is fundamental to the effective operation of spacecraft it is entirely appropriate that NASA continue to maintain a range of propulsion research and technology (R&T) programs including an R&T program on electric propulsion. The objective of this program is to develop the technology for electric propulsion systems with performance, lifetime, reliability, and interface characteristics required for a broad range of

national missions. Within this program is an effort to develop a fundamental understanding in the areas of plasmadynamics, ion physics, and improved materials and designs. The emphasis of this program is on high payoff technologies such as resistojets, arcjets, ion engines, and magnetoplasmadynamic (MPD) thrusters (Stone and Bennett 1989).

The NASA R&T program on electric propulsion is principally being carried out through two NASA centers: the Jet Propulsion Laboratory (JPL) and Lewis Research Center (LeRC). Both centers are working on a complementary program to advance the technology for electric propulsion systems for a broad range of future space platforms and spacecraft. JPL has been focusing principally on (1) improving ion engine life; (2) simplifying multi-engine system architecture; and (3) developing the technology necessary to build and test an MPD thruster capable of steady-state operation at power levels of several megawatts. LeRC has been focusing principally on (1) research on high-performance arcjets; (2) the basic physics and scalability of inert-gas ion thrusters and high-power MPD thrusters; and (3) technologies and studies essential for the integration of electric propulsion with space systems. There is a close interchange between JPL and LeRC, and the two programs both complement and supplement each other. Additional work is proceeding at Princeton University on MPD thrusters, at Michigan State University and Ohio State University on microwave electrothermal thrusters (MET), at Rocket Research, Inc., TRW and the University of Toledo on arcjets and at Hughes on ion engines.

This paper focuses primarily on progress beyond that reported at the 25th Joint Propulsion Conference (Stone and Bennett 1989). For references to the earlier work, the reader is referred to Stone and Bennett 1989, Stone 1988a, Stone et al. 1988b and King et al. 1989. An overview of NASA's advanced propulsion concepts program, including electrodeless thrusters, may be found in Bennett and Stone 1989. This paper is organized into two principal sections: one covering the ongoing base R&T program and the other covering the planned exploration technology program on nuclear electric propulsion (NEP).

RESEARCH AND TECHNOLOGY PROGRAM

The ongoing base R&T program includes electrothermal thrusters (resistojets and arcjets), electrostatic thrusters (ion engines), electromagnetic thrusters (MPD thrusters), and electrodeless thrusters. The following subsections summarize the latest NASA-sponsored developments in these areas.

ELECTROTHERMAL

Resistojets offer the dual advantages of specific impulses at or beyond traditional onboard chemical propulsion and the ability to utilize a range of propellants. Sub-kilowatt resistojets have already been flown to provide NSSK on GEO satellites (Sackheim and Howell 1984 and Byers and Wasel 1987). The continuing successes in the kilowatt-class arcjet program will allow their utilization to extend the lifetimes of GEO satellites. Moreover, arcjets can reduce spacecraft mass sufficiently to permit the use of smaller, cheaper launch vehicles with more flexibility on choice of launch sites.

Resistojets

Hydrazine resistojets are operational on INTELSAT V, RCA SATCOM, G-Star, and Spacenet communications satellites. Thrust levels range from about 0.2 N to 0.4 N with average exhaust velocities on the order of 2.9 km/s for electrical power inputs of about 0.3 kW to 0.5 kW. This represents a 30 percent increase in performance over that available from conventional hydrazine thrusters which translates into reduced propellant requirement or increased satellite life (Dowdy 1990).

LeRC is studying several resistojets, including a hydrazine resistojet thruster capable of a specific impulse of about 3100 m/s (320 lbf/lbm), multipropellant resistojets suitable for

use on Space Station Freedom (SSF), and water resistojets which provide for ease of and safety in handling.

The hydrazine resistojet program is in the middle of a four-phase contracted industrial development program to define, fabricate and characterize the thruster. Improved heat exchanger performance, longer-life, higher-temperature heaters, and improved nozzle performance are being sought (Stone and Bennett 1989).

Engineering model multipropellant resistojets have been fabricated by Rocketdyne and Technion, Inc. and successfully tested at LeRC. These resistojets offer the benefit to SSF of being able to safely use waste gases for propulsion thereby enhancing SSF control and minimizing waste disposal problems. The plan is to use SSF electrical power to heat an electric heat exchanger which will expel the SSF waste gas propellant with a high velocity (1 km/s to 5 km/s) depending on the gas and the gas temperature. Table 1 summarizes the design parameters and demonstrated results (Iacabucci et al. 1989).

Table 1. Engineering Model Resistojet Design Parameters and Demonstrated Results

<u>Parameter</u>	<u>Design</u>	<u>Demonstrated</u>
Life (h)	10,000	9,600 (135 thermal cycles)
Propellants	CO ₂ , N ₂ , He, H ₂ , O ₂ , Ar, Kr, Steam	CO ₂ , N ₂ , Ar Ar/N ₂ mixture
Specific impulse (m/s)	1,300 m/s (CO ₂ at 1573 K)	1,350 (CO ₂ at 1473 K)
	4,900 m/s (H ₂ at 1573 K)	5,300 m/s (H ₂ at 1473 K)
Thrust (N)	0.13 - 0.44	0.05 - 0.45
Thrust Chamber Pressure (kPa)	275.8	13.7 - 310.0
Maximum Operating Temperature (K)	1673	1573

Water resistojets are being developed for possible application to human-tended platforms such as the proposed industrial space facility (ISF). In order to operate successfully in space, a zero-gravity steam generator is needed to overcome stability and phase separation problems. Attention has been focused on a cyclone heater concept in which the liquid water is swirled to the outer wall and boiled by heat radiated from the central heater which also heats the resistojet by conduction. Laboratory models have been built and tested. In an attempt to determine the sensitivity of the two-phase water vaporizer the water resistojet was rotated about the horizontal axis normal to the thruster axis as shown in Figure 2. Liquid pooling was observed in the bottom of the boiler chamber. Ultimately, a flight test will be needed for validation.

Arcjets

A major program being initiated by NASA is the Earth Observing System (EOS) which will aid in developing a better understanding of the overall global environment of Earth. EOS spacecraft, which will be in polar orbits at altitudes of about 700 km, may have masses on the

order of 13,000 kg. EOS mission velocity increments (Δv) are presently estimated to be ~164 m/s for orbit acquisition, ~75 m/s for 7.5-year orbit maintenance, and ~232 m/s for safe disposal. Since about 30% of the EOS-A wet mass is power, propellant (monomethyl hydrazine, hydrazine and nitrogen tetroxide) and propulsion and 26% is payload it is clear that a factor of two reduction in propulsion and power mass would add 50% more payload to EOS-A.

Research being performed by LeRC, Rocket Research, Inc., and others has shown that arcjets offer a factor of 1.5 to 2.5 increase in specific impulse over chemical propulsion systems. In the case of EOS-A, an LeRC analysis showed that using a 1.5-kW arcjet with a LeRC-developed high-temperature chemical thruster would allow use of a common propellant feed system and elimination of the monomethyl hydrazine. This in turn would yield (1) an exhaust which contains no carbon products; (2) simplification of the propulsion system design; and (3) higher specific impulse and mass savings. The total mass saving for EOS-A was calculated to be 394 kg (Sovey et al. 1990). At a time when experiments are being evaluated for possible removal because of power and mass constraints a 394-kg saving is important.

Perhaps the most dramatic recent breakthrough in electric propulsion occurred in 1989 with AT&T's award of a contract to GE Astro Space to build the Telstar 4 satellites using Rocket Research arcjets for stationkeeping. Reportedly the use of the arcjets will extend the life of each satellite up to four years and reduce the mass of the spacecraft thereby allowing the use of lower cost launch vehicles (Isbell 1989 and Marcus 1990). With GEO satellites (such as Telstar 4 and the Tracking and Data Relay Satellite, TDRS) having effective mission velocity increments in the range of 2 km/s the mass savings associated with higher performance apogee and on-orbit thrusters become very important in reducing overall mass and/or increasing the payload and operational life (Sovey et al. 1990).

The selection of arcjets for Telstar 4 and their consideration for other spacecraft represents the culmination of a six-year research program by LeRC, Rocket Research, GE, and TRW. This program has

- Completed a 1000-hour/500-cycle, 1.3-kW arcjet life test
- Met the thermal/mechanical interfaces
- Successfully designed and tested a power processor
- Addressed the integration issues (such as plumes)

Rocket Research, Inc. has successfully completed system-level thermomechanical (vibration) environmental qualification of the 1.3-kW arcjet. In order to overcome gassifier nonvolatile residuals, a dual inlet (injector) design was demonstrated as shown in Figure 3. A 900-hour/900-cycle test was successfully completed with this design.

An obvious concern with arcjets is the radiated electromagnetic interference (EMI) from the arc discharge and the plume. Tests are under way at TRW and LeRC to evaluate radiated EMI. Preliminary results (see Figure 4) indicate low levels of EMI in the communication frequency ranges with an as yet unexplained slow increase with decreasing frequency.

As a further aid in assessing the potential impacts of arcjet plumes, LeRC has assembled and achieved initial operational capability (IOC) on an arcjet plume facility which will permit near- and far-field plume analyses using a high-speed positioning system with electrostatic probes and optical diagnostics (Manzella et al. 1990). Near-field plume plasma properties (electron number density and temperature as a function of position and species identification and distributions) have been measured. The results of this investigation indicate that the arcjet plume is a weakly ionized, highly non-equilibrium plasma. Species measured in the plume include molecular nitrogen, atomic hydrogen, molecular nitrogen ions, and NH which indicates

some recombination occurs within the nozzle.

Another key issue is developing the integration technology for kilowatt-class arcjets and power processing units (PPUs). Data obtained by LeRC using an injected AC signal to generate a map of arcjet plasma versus frequency indicate that the load characteristics depend on power level, propellant mass flow rate and environmental test conditions (Hamley 1990). A mathematical model describing the dynamic impedance in the frequency range studied (100 Hz to 100 kHz) was developed from these data. This model will be useful in the design of future PPUs.

Electrodes are obviously the critical components in arcjet operation. LeRC has evaluated cathode materials for startup and life performance. The results of this evaluation have shown that low work function additives are essential for reliable startups but are probably not essential to steady-state emission or lifetime. The steady-state emission appears to be a combination of field and thermionic emission and comes from a molten crater in the cathode. Based on tests of over 1000 hours with no degradation the tungsten-based cathode appears optimal for lifetime.

Tests with a segmented arcjet anode were continued in order to study the impacts of nozzle design and plasma attachment on arcjet performance (Curran et al. 1990). As illustrated in Figure 5, the high mass flow rates forced the arc to attach farther downstream than occurred under low flow rates but the current distribution was not significantly affected by the current level. To date, the overall performance has not been found to be very sensitive to arcjet nozzle configuration which implies that the cathode physics dominate performance expectations.

An exciting aspect of arcjet technology is to extend development to power ranges above and below the nominal 1-kW level. As a first step in this direction, LeRC has designed, fabricated, and evaluated a 5-kW arcjet full-bridge, pulse-width-modulated power converter achieving an efficiency of 93.5% at 5 kW and 50 A (Gruber et al. 1989). A 1-kW-class arcjet system was successfully tested to 1.8 kW and work is proceeding to develop a 2- to 5-kW-class arcjet.

At the low power end, LeRC has achieved stable, reliable operation below 0.5 kW with a number of arcjet anodes and mass flow rates (Curran and Sarmiento 1990). Figure 6 shows the specific impulse as a function of specific power for the highest and lowest mass flow rates used in the experiments. As the mass flow rate decreases there is an indication of an increasing loss mechanism that may be a viscous effect or the result of increasing frozen flow losses. Nevertheless, the results indicate that it should be possible to develop a stable, low-power arcjet to replace the lower specific impulse chemical thrusters currently being used for NSSK on GEO satellites.

Under the sponsorship of the Strategic Defense Initiative Organization (SDIO) and NASA, JPL conducted a study of 30-kW ammonia-propellant arcjets. This work, which extended over 3.5 years and led to the design of an arcjet with improved performance and thermal characteristics using a space storable propellant, has recently been summarized in Deininger et al. 1990. System studies reported by Deininger et al. 1990 have shown that the use of 30-kW arcjet propulsion upper stages for deployment and maneuvering can significantly reduce the number of launches and deployment times compared to those achievable with advanced chemical upper stages. In this program duration tests up to 413 hours were run. Arcjet engine performance was found to be relatively insensitive to cathode tip geometry, with the cathode tip erosion rate increasing with increasing arc current. As the cathode tip erodes a maximum distance may be achieved beyond which the arc may pull back from the anode nozzle into the constrictor, likely leading to destruction of the thruster. While this work indicates that the 30-kW ammonia arcjet could be developed into a flight design, Deininger et al. 1990 have identified the need for technical improvements including higher efficiency, greater specific impulse, and longer lifetime.

ELECTROSTATIC

A recent readiness appraisal concluded that "ion propulsion stands on a substantial, twenty-five year base of theoretical, technological and flight system development. Mature flight hardware, mostly fabricated using production methods and tested using sophisticated techniques, has evolved from this background and is available from a number of sources . . . Thus, it is concluded, given existing data and experience, that ion propulsion is ready for the next generation of communication satellites" (Schreib 1988). There is a substantial body of literature that shows the benefits (reduced mass, smaller/cheaper launch vehicles, longer lifetime, larger payloads) to be achieved with ion propulsion (in addition to the references in the survey papers already cited the interested reader is referred to Patterson and Curran 1989 and Garner 1989a). Clearly the high efficiency and specific impulse of ion engines makes them a serious contender for future Earth-orbiting and planetary spacecraft.

Ion propulsion systems in the power range of 5- to 10-kW appear to be the most likely candidates for near-term application of ion propulsion and the ongoing NASA work in this power regime will lay the foundation for higher power ion engines in the tens to hundreds of kilowatts needed for future space exploration (Byers 1989 and Bennett 1984). With the ongoing development of advanced space power systems there is a growing opportunity for the use of ion propulsion. [Note: A discussion of the NASA advanced space power programs may be found in Bennett 1989.] Consequently, the NASA ion propulsion program at JPL and LeRC is focusing on this power range but with a longer term view of achieving even higher powers. Both JPL and LeRC are studying ion engines operating with inert gases (e.g., xenon) as propellants instead of mercury as was used in earlier experiments because these inert propellants are nontoxic while having properties that provide the desired performance and simplified flow control.

The main focus of both the JPL and LeRC work is on increasing both the operating power level and the operating life. A principal limitation on ion engine operating life is ion sputter erosion of critical engine components such as the upstream face of the screen grid, the downstream face of the accelerator grid, and the baffle (used on the J-series ion engines) (Garner et al. 1989b and Rawlin 1988). While low, the ion sputter rates can, when integrated over the relatively long operating times required for ion engines, remove significant portions of the component (Garner and Brophy 1990).

Earlier studies at JPL have shown that the addition of small quantities of nitrogen to the xenon propellant may reduce the erosion by up to eighteen-fold in the materials tested. As shown in Figure 7, a fifteen-fold reduction in the erosion rate of the cathode side of the tantalum baffle was achieved in a J-series ion engine (Garner et al. 1987 and Garner and Brophy 1990).

LeRC-sponsored studies confirmed that the source of the high baffle erosion was a cathode jet (Friedly 1990), as originally suggested by JPL (Brophy and Garner 1988). The LeRC approach to the problem of erosion of the discharge baffle was to eliminate the baffle by using a ring-cusp engine design that uses a strong magnetic field for focusing. After a 900-hour test of a 5.5-kW ring-cusp ion thruster in which the discharge igniter experienced significant erosion, LeRC has developed a new operational baseline in which discharge is achieved without the use of an igniter. LeRC intends to continue to establish a 10,000-hour capability for the 5.5-kW ion engine. It is clear from the 900-hour test that the facility, through residual background gases, can degrade the ion engine, specifically the accelerator grid. A facility with increased pumping speeds is clearly needed.

In an effort to reduce development costs and system complexity, JPL has been studying the ion engine throttling strategy. As shown in Figure 8, using a 3000-kg spacecraft mass lunar rover mission with a 30-kW xenon ion propulsion system as a baseline, JPL has shown that throttling at a constant beam current eliminates the need for active propellant flow controllers and complex engine throttling software (Garner et al. 1988). Future plans include operating the engines on the simplified throttling strategy over a power range of 2:1 and developing

computer algorithms for autonomous control of the simplified throttling strategy.

Differential thermal expansion can lead to distortions of the ion engine grid; therefore, models are under development to predict grid behavior under load (MacRae and Hering 1990). In order to measure this differential displacement JPL has developed an optical technique using laser imaging (the type II confocal optical microscope technique) and is preparing to conduct nonintrusive measurements (Trava-Airoldi et al. 1990).

In order to reduce the cost of long-term testing of ion engines at fixed design points and to expedite determination of the optimized lifetime/performance tradeoff, LeRC has developed a facility which makes in situ, real-time measurements of wear rates. Figure 9 shows a schematic of the experimental setup for measuring the atom density of molybdenum using a laser-induced fluorescence (LIF) technique.

ELECTROMAGNETIC

The NASA-sponsored research on electromagnetic propulsion is focused primarily on magnetoplasmadynamic (MPD) thrusters. The MPD thruster combines high specific impulse (10,000 to 50,000 m/s) with moderate thrust levels (10 N to 100 N) which makes it a candidate for robotic cargo vehicles and for piloted spacecraft. The attractive feature of the MPD thruster compared to the arcjet and the ion engine is its potential to be able to process megawatts of power. The NASA-sponsored research is aimed at increasing the power, efficiency and lifetime of MPD thrusters. Mission applications involving MPD thrusters will probably require that the thrusters have efficiencies greater than 50% with lifetimes greater than 1000 hours.

JPL is approaching MPD performance improvement through advanced electromagnetic design, thermal design and mechanical design. A buffer electrode design for a radiation-cooled, self-field MPD thruster, as shown in Figure 10, was designed and built as part of this effort. However, the results from 20 steady-state runs (4 hours accumulated run time) at powers up to 72 kW showed the buffer electrode worsened performance.

JPL has gone on to design a radiation-cooled, applied-field MPD thruster as shown schematically in Figure 11 based on an earlier AVCO design (Pivrotto and Goodfellow 1990). JPL is completing work on a thrust stand for MPD thrusters. LeRC has also fabricated a new azimuthal field MPD thruster and will test it when facilities are cleared for operation.

LeRC has completed its 250-kW, steady-state MPD thruster test facility and assembled several MPD thrusters with various electrode and applied field geometries. Preliminary results from testing conducted at power levels between 50 kW and 112 kW show a three-fold improvement both in efficiency (from 7% to 21%) and in specific impulse (from about 6400 m/s to 15,700 m/s) with increasing applied field strength between 0 and 0.4T (Myers et al. 1990). In tests at power levels up to 8 kW LeRC has found that the concept of hollow cathodes in MPD thrusters appears promising as a way to improve performance (e.g., higher currents and temperatures) without incurring damage to the cathode. A schematic of such a device along with the preliminary performance is shown in Figure 12 (Mantenieks 1990).

As part of the ongoing effort to understand MPD thruster dynamics and to guide improved designs and operation, both JPL and LeRC are developing MPD computer models. JPL is developing a cathode sheath model and an electrode heat-transfer model in order to quantify the phenomena associated with cathode tip heating. The preliminary results on cathode heat load are compatible with the measured average axial temperature gradient. LeRC has developed a steady-state, one-fluid, cylindrical-geometry model of a self-field MPD thruster and is running test cases. Applied field effects will be incorporated in the near future. A comparison of the LeRC model and one developed at the Institut fur Raumfahrtssysteme (IRS) at Universitat Stuttgart is shown in Figure 13.

The Electric Propulsion Laboratory (EPL) at Princeton University is continuing its research on coaxial plasma thrusters (specifically MPD thrusters), including modeling plasma wave phenomena and studying (1) anode processes, (2) acceleration mechanisms, and (3) erosion processes. An analytical study has been completed on mass savings potentially realizable with MPD propulsion for LEO to GEO transfer. This study defines the realm of mass and cost savings. EPL continues to improve its diagnostic capabilities including an optical diagnostic system to investigate the plasma state properties in the interelectrode region and triple probe measurements in the vicinity of the anode lip and in the plume (EPL 1990).

LeRC-sponsored research on MPD magnetic nozzles is under way at Ohio State University. The effects of an applied coaxial magnetic field have been measured on a pulsed MPD thruster (York 1990).

ELECTRODELESS THRUSTERS

One very direct method of overcoming electrode erosion is to eliminate the electrodes. Such electrodeless thrusters should be useful for a number of different kinds of space missions and they offer the potential of greater flexibility in the choice of propellants. Both JPL and LeRC are conducting research on electrodeless thrusters. JPL has been investigating the electron-cyclotron-resonance (ECR) plasma engine and LeRC has fabricated a high-power microwave electrothermal thruster (MET) concept.

The ECR, shown schematically in Figure 14, uses microwave power to create a plasma which is accelerated through a diverging magnetic field. JPL has built and operated an experimental test bed in concert with developing computer models of the plasma trajectory and of the nonequilibrium and radiation effects. The JPL measurements of argon propellant utilization, ion energy, plasma potential, and plasma beta are consistent with theoretical predictions based on measured electron temperature. Work is continuing to understand why the electron temperatures have been lower than predicted by theory (Culick and Sercel 1990).

The LeRC MET concept, shown schematically in Figure 15, uses 915-MHz continuous wave (CW) microwave power to create a plasma discharge in the propellant that in turn heats the propellant so it expands through a throat-nozzle assembly. A low-temperature superconducting 5.7-T magnet will be used to create a magnetic nozzle to compress the ionized propellant along the thruster axis. Nitrogen, helium and hydrogen will be tested as propellants in the MET, at discharge chamber pressures to 1 MPa (Power and Chapman 1989).

EXPLORATION TECHNOLOGY PROGRAM

In a speech on 20 July 1989, commemorating the 20th anniversary of the Apollo 11 lunar landing, President Bush established the basic framework for future exploration planning. He stated: "First for the coming decade -- for the 1990s -- Space Station Freedom -- the critical next step in all our space endeavors. And next -- for the new century -- back to the Moon. Back to the future. And this time, back to stay. And then -- a journey into tomorrow -- a journey to another planet -- a manned mission to Mars." On 2 November 1989, the President approved a national space policy that reaffirmed that a long-range goal of the civil space program is to "expand human presence and activity beyond Earth orbit into the solar system". As part of the Fiscal Year (FY) 1991 budget, the Bush Administration strongly endorsed the initiation of the Space Exploration Initiative (SEI), a focused, multi-decade program of human exploration of the Moon and Mars. In a speech on 11 May 1990, President Bush expressed a desire to have astronauts on Mars by the time (2019) of the 50th anniversary of the Apollo 11 landing.

Following the President's speech on 20 July 1990, NASA undertook a 90-day study of the human exploration of the Moon and Mars. This study noted that "Space is an infinite source of challenges. To send humans to explore it and use its resources, development must begin today of new technologies in many areas, including . . . nuclear propulsion". The study went on to note that "Detailed mission planning for human exploration has centered on the use of high-performance cryogenic engines and aerobraking technologies as the foundation for Earth-Moon and Earth-Mars transportation. However, nuclear propulsion is a major alternative for Earth-Mars transportation. Two specific technologies are nuclear thermal rockets and nuclear electric propulsion" (NASA 1989).

The NASA 90-day study report observed that "Nuclear electric propulsion offers very high specific impulse (2,000 - 10,000 seconds) which results in a reduction in propellant mass requirements. By nature, low thrust electric propulsion systems offer increased mission flexibility compared to high thrust chemical propulsion systems (i.e., unconstrained by launch windows). At power levels of 5 megawatts, nuclear electric propulsion has the potential to reduce initial mass to low Earth orbit for Mars cargo missions by 50 - 60% compared to chemical/aerobrake systems. For the cargo missions, trip time to Mars will be relatively long compared to chemical propulsion systems. At considerably higher power levels (40 - 200 megawatts), nuclear electric propulsion can potentially reduce trip time for piloted mission applications.

"Megawatt level nuclear power systems with low specific mass will be required. The SP-100 reactor development program will demonstrate space reactor technology and can potentially be used for the 5 megawatt class systems when coupled with advanced dynamic energy conversion systems. For the higher power nuclear electric propulsion concepts (40 to 200 megawatts), a new reactor development program will be required, as well as development of energy conversion, heat rejection, and power management and distribution techniques to achieve low power system specific masses" (NASA 1989).

Subsequent to the release of the 90-day study, the Vice President requested that the National Research Council (NRC) assess the scope and content of the NASA document as well as alternative approaches and various technology issues. The NRC Committee on Human Exploration of Space conducted the requested assessment, which included among its recommendations (NRC 1990):

"Consideration should be given to demonstration of the nuclear electric power system as the power source for an electric propulsion system, which may have application to science missions with large launch velocity requirements. (In fact, a number of outer planet missions have been suggested, including a Jovian system grand tour, that will require such advanced power sources.) Here, as with the nuclear rocket, considerations of safety must be incorporated into research, development, and demonstrations and factored into assessments of overall systems performance. The nuclear electric system might be demonstrated within these constraints by a mission in which the system is launched to a high orbit, say 600 miles, before it is operated. The orbit could then be raised by nuclear-electric propulsion to geosynchronous orbit or beyond.

"If safety concerns can be successfully addressed, and feasibility demonstrated, the committee believes that use of nuclear power and propulsion can meet many needs in the human exploration of space."

These recommendations are well founded on earlier studies, such as NASA 1988 which identified nuclear electric propulsion (NEP) as a critical pacing element in going to Mars and NCOS 1986 which stated:

"A more efficient, lower-thrust vehicle is needed to supplement the chemical vehicle for the long-range transportation of bulk cargo such as propellants. Electric propulsion is ideal

for this application. Electric propulsion is needed for high energy robotic missions to asteroids, comets, and the fascinating worlds of the outer Solar System.

"Solar-powered and nuclear-powered electric vehicles are needed to launch robotic outer Solar System exploration missions. With modular basic thrusters the vehicle can use either power source. Saturn ring rendezvous and probing the Uranian Ocean will require such a capability. These electric vehicles will also become the workhorses that provide logistic support to Martian operations . . . We recommend that: **Electric powered transfer vehicles be developed suitable for either solar or nuclear power.**" (Emphasis in the original.)

In laying the foundation for the research and technology program needed for the kind of space exploration identified in the foregoing cited report of the National Commission on Space, a technology benefits assessment found that "A high power electric thruster capable of delivering over 3,000 seconds of specific impulse is a candidate propulsion system for a Mars cargo vehicle; electric propulsion may also be applied in advanced robotic exploration missions. This technology will substantially reduce the amount of propellant needed, and the time-in-flight required for deep space missions. For example, electric propulsion would allow the mass required in Earth orbit for a piloted Mars mission to be reduced by 400,000 pounds, eliminating the need for two 200,000 pound heavy lift launch vehicles. Two electric propulsion technologies are currently considered candidates for applications, ion and magnetoplasma-dynamic (MPD) . . . MPD systems, have the advantage of being simple and compact, but only limited research has been done because they require megawatt-level power to perform effectively. Ion systems have been operated at reasonable efficiency for over a decade, but substantial improvements are still needed. Long-term testing of both technologies is needed" (NASA 1987).

In a recent evaluation of a variety of advanced low-thrust propulsion options for the cargo-delivery portion of a split-mission piloted Mars exploration scenario, it was concluded that "A 100-MWe class NEP system has superior mass and trip-time performance. Only advanced solar sails can achieve a lower IMLEO [IMLEO = initial mass into low Earth orbit], yet the NEP has a shorter trip time than the chemical system. In fact, the NEP system can make a round-trip in less time, and with a lower IMLEO, than a chemical system making a one-way delivery" (Frisbee et al. 1989).

The NEP spacecraft concept considered in Frisbee et al. 1989 is shown in Figure 16 and the NEP mission scenario is shown in Figure 17. The performance of the 100-MWe-class NEP system is illustrated in Figures 18 and 19 for various combinations of power and specific impulse. As Frisbee et al. 1989 observe: "The most striking result is the performance of the high- I_{sp} (20,000 $lb_f\text{-s}/lb_m$) NEP system at 100 MWe. This vehicle has an IMLEO (700 MT) that competes directly with solar sails, yet has an Earth-to-Mars trip time (240 days) that is less than that required for the minimum-energy trajectory chemical system. At an I_{sp} of 10,000 s, a 300-MWe system has an IMLEO comparable to the chemical system, but with less than one-half the trip time of the chemical system. Even at a modest I_{sp} (5,000 $lb_f\text{-s}/lb_m$), the NEP system can best the chemical system in trip time".

These results are generally consistent with an independent study sponsored by NASA's Langley Research Center (LaRC) which concluded: "The propulsive option which requires the lowest initial LEO weight is the hybrid NEP/CHEM option for flight times of 1.0-1.2 years, the dual vehicle CHEM/NEP option for flight times of 1.2-1.3 years, the NTP option for flight times of 1.3-1.7 years, and either a pure SEP or NEP vehicle option for flight times of 1.7-2.5 years" (Braun and Bliersch 1989).

As a part of the SEI, NASA has established an Exploration Technology Program which includes nuclear propulsion. The nuclear propulsion thrust is focused on developing the

technologies to satisfy the mission requirements being defined in ongoing NASA studies. In planning the nuclear propulsion thrust it is recognized that there are several competing concepts in both nuclear thermal propulsion (NTP) and nuclear electric propulsion (NEP). The thrust scope covers development of the total nuclear propulsion system and supporting technologies for piloted and cargo missions to Mars. In order to expedite the technology, the thrust plan defines a parallel, iterative, dual-path approach of concept development and technology development to be carried out over three phases:

- Phase 1 (~1991 - 1995): System Definitions
- Phase 2 (~1995 - 2000): System Development
- Phase 3 (~2000 - 2005): System Demonstration

The overall goals of the nuclear propulsion technology thrust are to

- Develop the technologies required to apply space nuclear propulsion systems to improve the mission performance for human missions to Mars
- Identify and develop at least one space nuclear propulsion system that, alone or in combination with other propulsion systems, meets the propulsion requirements of the piloted Mars mission and for which technical feasibility issues have been resolved

Specifically for NEP the project-level goal is to develop the NEP technologies, including nuclear reactor systems technologies, advanced low-mass radiator and power management systems, and high-power, long-life electric thrusters for piloted missions to Mars, including unmanned precursor missions. Key technical issues to be addressed include:

- Safety/safeguards/quality assurance during all program phases (design, development, test and engineering [DDT&E], fabrication, launch, assembly, checkout, operations)
- Qualification and acceptance test strategies
- Reliability and fault tolerance
- High performance thrusters (including space reactors)
- Reusability/restart capability
- Reactor fuel
- Structural aspects
- Turbomachinery/pumps/valves (advanced dynamic power conversion systems)
- Reactor vessels
- Diagnostic capability (health monitoring and propellant mass measurement)
- Control systems (including neutronics and control and instrumentation and control)
- Power processing units (PPUs)
- Space operations (including radiation shielding and design criteria for operation and maintenance of the nuclear propulsion system in space)
- Propellants and propellant handling
- Thermal hydraulics
- Thermal management
- Materials
- Lifetime
- Mass/volume limitations
- In-situ propellant utilization

As a first step in preparing for a planned FY 1991 program new start on nuclear propulsion, JPL hosted a NEP workshop in Pasadena on 19-22 June 1990 in cooperation with NASA, the Department of Energy (DOE), and the Department of Defense (DoD). The purpose of the workshop was to obtain from the experts ("concept focal points") state-of-the-art information on electric propulsion (EP) systems and reactor power sources in the context of

mission impact, developmental status, and programmatic requirements for many candidate technologies in order to develop the technical work scopes for FY 1991 and beyond. Technology review panels covering mission analysis, propulsion technology, reactor technology, advanced development planning, and safety reviewed the various EP and reactor systems, including

Electric Propulsion Candidates

- Steady-state MPD thruster
- Burst-mode MPD thruster
- Deflagration thruster
- Pulsed plasmoid thruster
- Pulsed electrothermal thruster
- Ion thruster
- ECR thruster
- Ion cyclotron resonance thruster
- Pulsed inductive thruster

Reactor Power Source Candidates

- SP-100 growth reactor
- Nuclear Rankine system
- NERVA-based reactor*
- Particle bed reactor
- Pellet bed reactor
- Thermionic reactors
- Gas core reactor

*NERVA = Nuclear Engine for
for Rocket Vehicle
Applications

Special presentations were also given on mission analysis, electric propulsion overview, power technology overview, safety, power beaming, thermal management technology, SP-100 space reactor mission analysis, coaxial plasma thrusters, and the Odyssey concept.

The information from this and the NTP workshop hosted by LeRC on 10-12 July 1990 will be evaluated and assembled in time for a September 1990 presentation to the NASA/DOE/DoD nuclear propulsion steering committee. From this it is anticipated that an NEP program will be developed.

CONCLUDING REMARKS

The NASA electric propulsion program continues to make excellent progress in developing the technology to provide significant benefits to a wide class of space missions. Some of this progress has begun to appear in planned space missions such as the planned use of resistojets on Space Station Freedom and the planned use of arcjets on Telstar 4. Both ion and MPD thruster technologies are being developed with the objective of having them ready for future missions such as a cargo vehicle flight to Mars.

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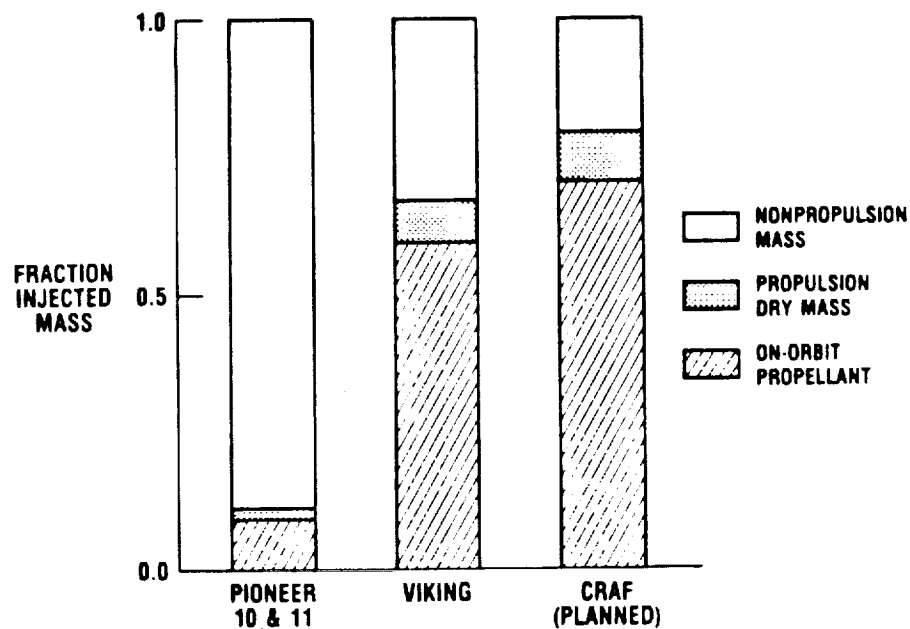


Figure 1. Planetary spacecraft injected mass fractions.

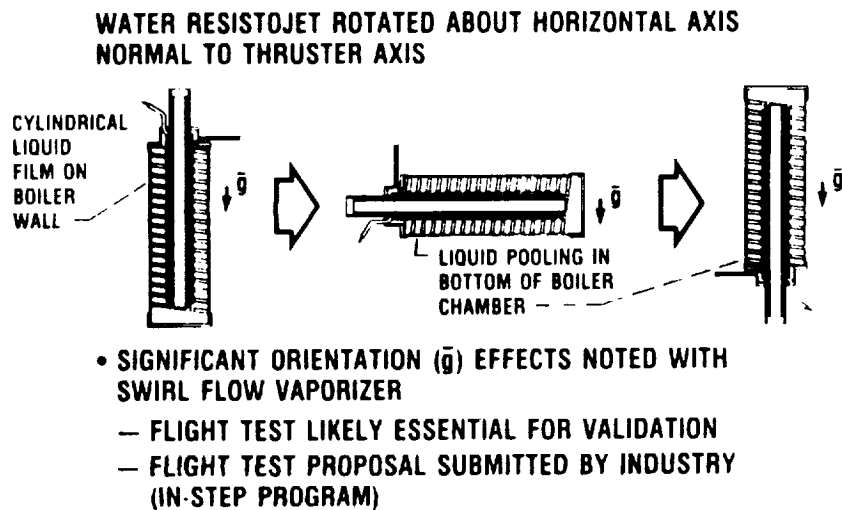
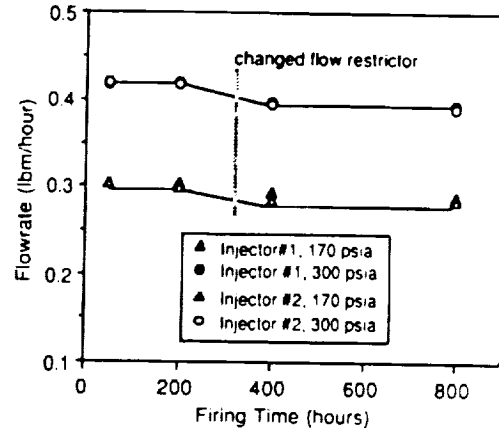
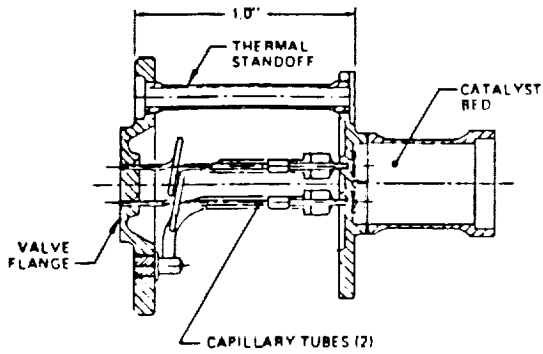


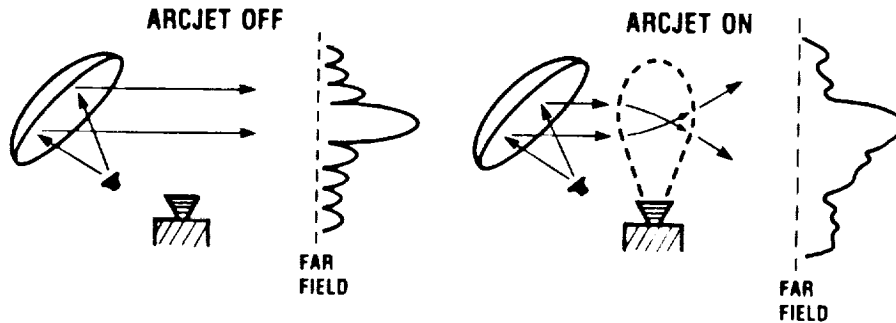
Figure 2. Evaluation of a water/waste gas resistojet with two-phase water vaporizer

0 DUAL INLET (INJECTOR) DESIGN
 0 AUTOMATED, CYCLIC TEST ON ARCJET SIMULATOR
 0 WORST CASE BLOWDOWN CONDITIONS - EACH SIDE



0 DUAL INLET DESIGN SOLVES NVR PROBLEM
 0 900 HOUR/900 CYCLE TEST SUCCESSFULLY COMPLETED
 0 SYSTEM REQUIRES VALVE DEVELOPMENT

Figure 3. Life test of a long-life gas generator (GG) for an arcjet.



- IMPACTS ON SIGNAL AMPLITUDE AND PHASE DETERMINED
 - NEAR FIELD/FAR FIELD
 - VERSUS PLUME DENSITY
- APPEARS BENIGN FOR 1 kW CLASS ARCJET SYSTEMS FOR REALISTIC S/C CONFIGURATIONS

Figure 4. Evaluation of arcjet communications impacts.

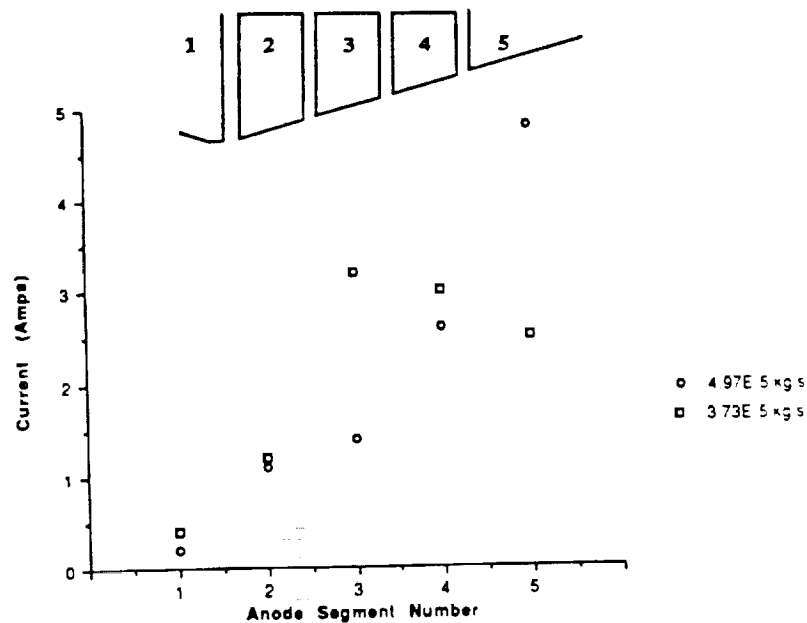


Figure 5. The effect of mass flow rate on current distribution in an arcjet with a segmented anode.

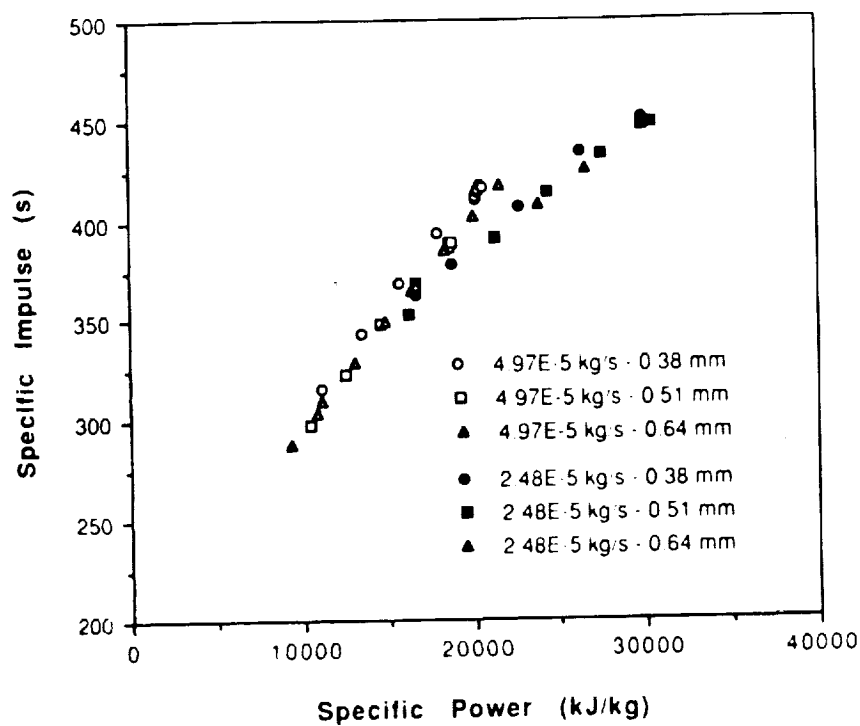
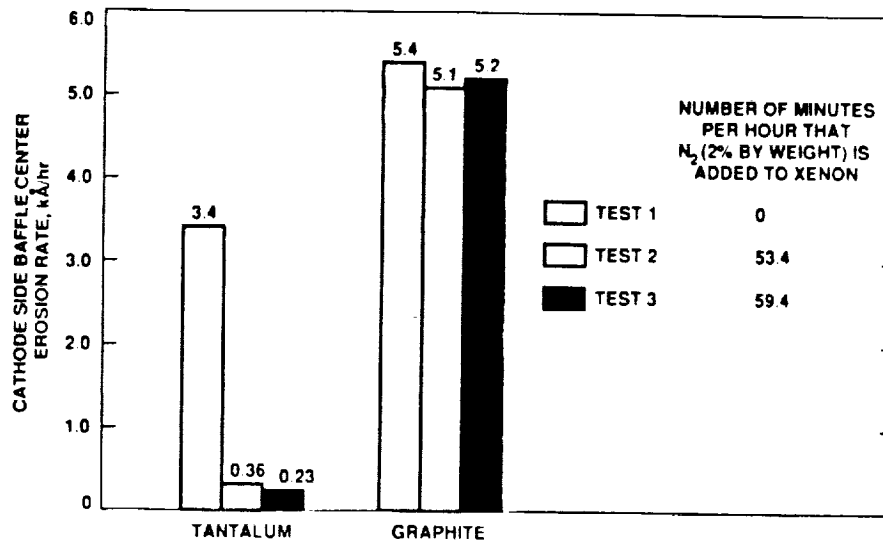


Figure 6. Specific impulse as a function of specific power for the highest and lowest mass flow rates used in the low-power arcjet experiments.

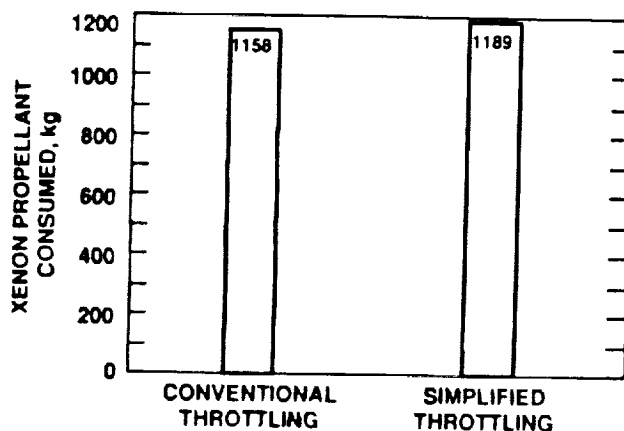
- THE ADDITION OF NITROGEN (2% BY WEIGHT) TO XENON PROPELLANT REDUCES BAFFLE EROSION BY A FACTOR OF 15



TANTALUM AND GRAPHITE EROSION SITES (AT BAFFLE CENTER) ARE EXPOSED TO IDENTICAL PLASMA CONDITIONS

Figure 7. Results of studies of discharge chamber erosion in an ion thruster showing the benefits of adding nitrogen to the xenon propellant.

- THROTTLING STRATEGY IMPACTS ON LUNAR ROVER MISSION WERE ASSESSED. A 30 kW XENON ION PROPULSION SYSTEM AND A 3000 kg SPACECRAFT MASS WERE ASSUMED



- ONLY 3% MORE PROPELLANT REQUIRED BY SIMPLIFIED STRATEGY
- NO TRIP TIME IMPACT

Figure 8. Results of a study of throttling on a xenon ion propulsion system.

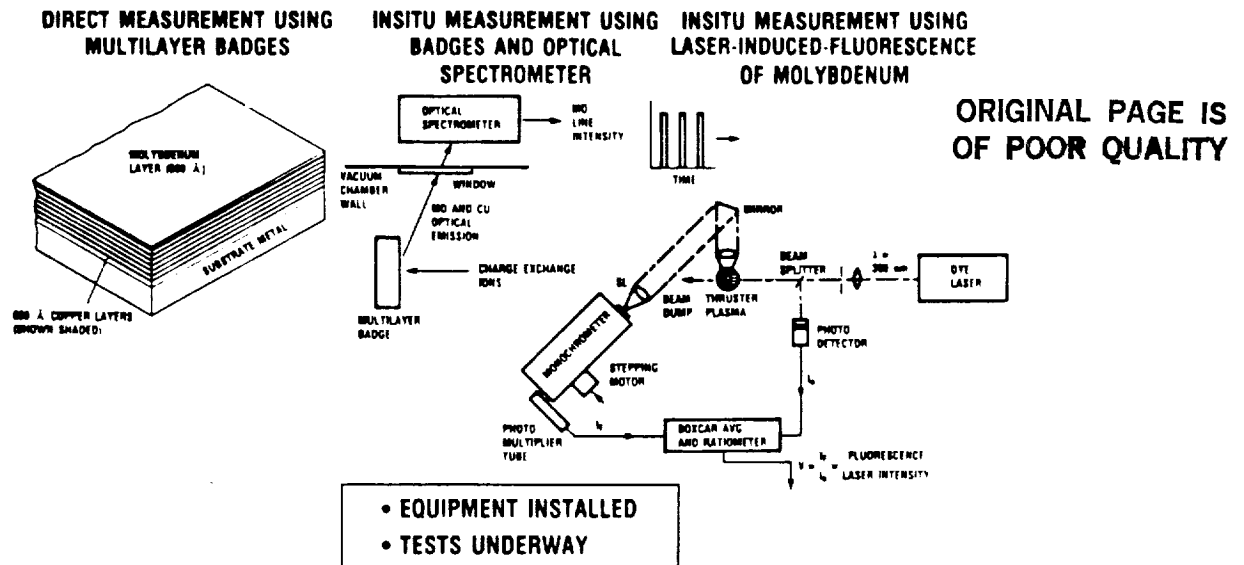


Figure 9. Schematic of a laser-induced-fluorescence (LIF) facility for making in situ, real-time measurements of wear rates of molybdenum for ion engine studies.

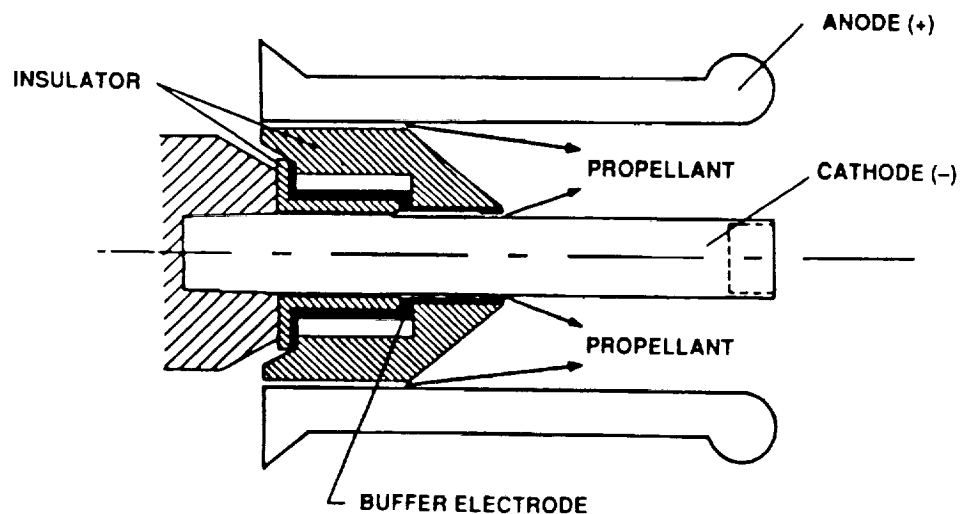


Figure 10. Schematic of a radiation-cooled, self-field magnetoplasmadynamic (MPD) thruster with a buffer electrode in an open throat cylindrical design.

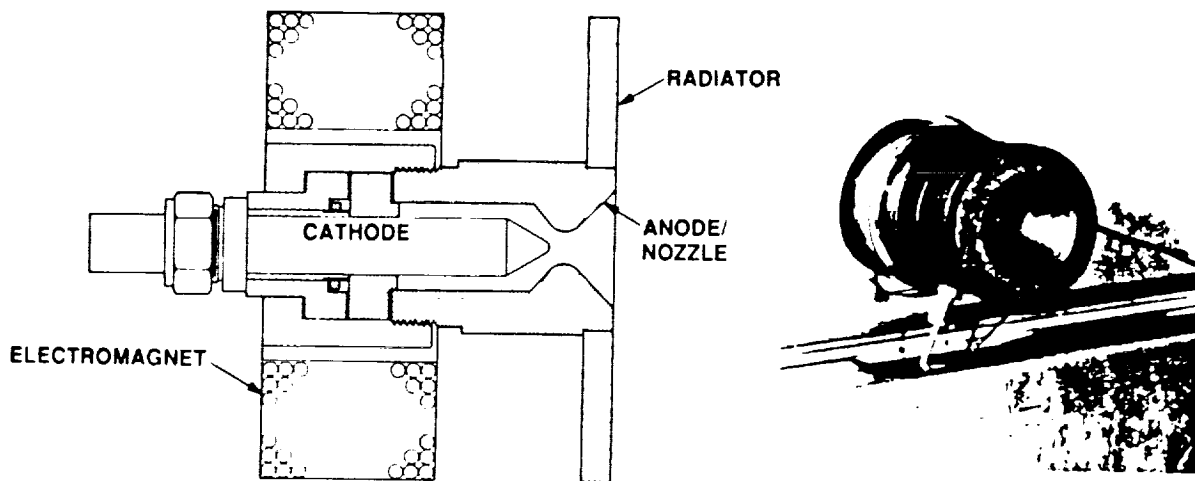


Figure 11. Schematic of a steady-state, radiation-cooled, applied field MPD thruster. A pure tungsten nozzle/anode and molybdenum body have been fabricated.

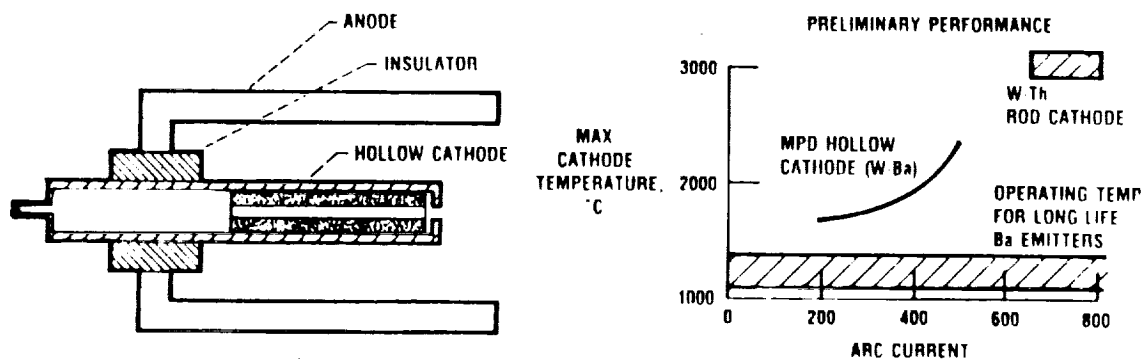
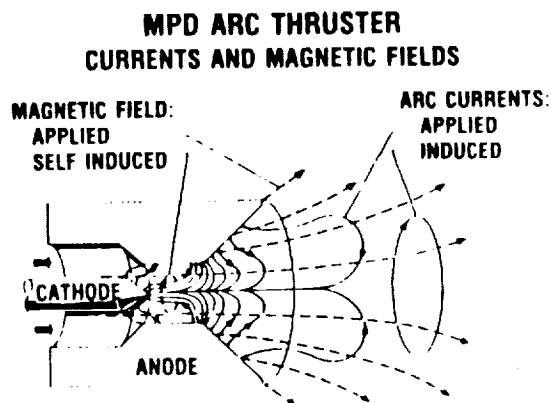


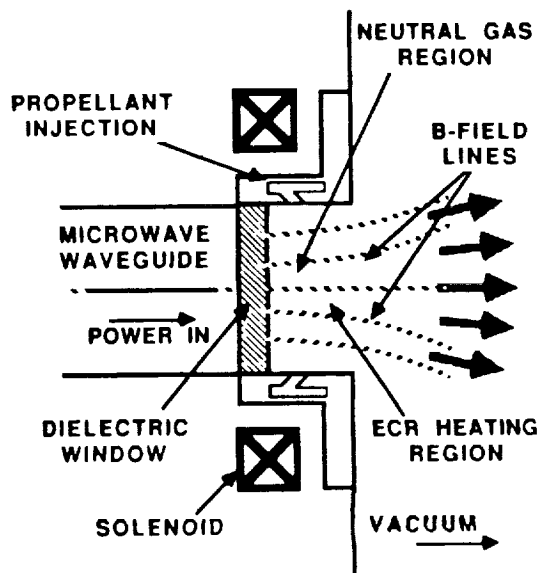
Figure 12. Schematic of a hollow cathode concept for an MPD thruster and the preliminary performance.



STUTT GART/IRS	NASA LeRC
TIME DEPENDENT	STEADY STATE
INDUCED FIELD	INDUCED AND APPLIED FIELDS
INVISCID FLOW	VISCOUS FLOW

STATUS
<ul style="list-style-type: none"> • LOW POWER SIMULATIONS OF CYLINDRICAL MPD THRUSTERS • PRELIMINARY SIMULATIONS OF > 100 kW MPD THRUSTERS • FUTURE SIMULATIONS TO INCLUDE IONIZED PLUME, COMPLEX GEOMETRIES

Figure 13. Comparison of the MPD models developed by Lewis Research Center (LeRC) and the Institut fur Raumfahrtsysteme (IRS) at Universitat Stuttgart.



CONCEPT:

MICROWAVE POWER IS USED TO CREATE A PLASMA WHICH IS ACCELERATED THROUGH A DIVERGING MAGNETIC FIELD

BENEFITS:

ELECTRODELESS DESIGN MAY ENABLE

- SCALABILITY FROM KW TO MW
- LONG LIFE
- HIGH EFFICIENCY
- CHOICE OF PROPELLANTS (e.g. lunar oxygen)

Figure 14. Schematic of the electron-cyclotron-resonance (ECR) plasma engine.

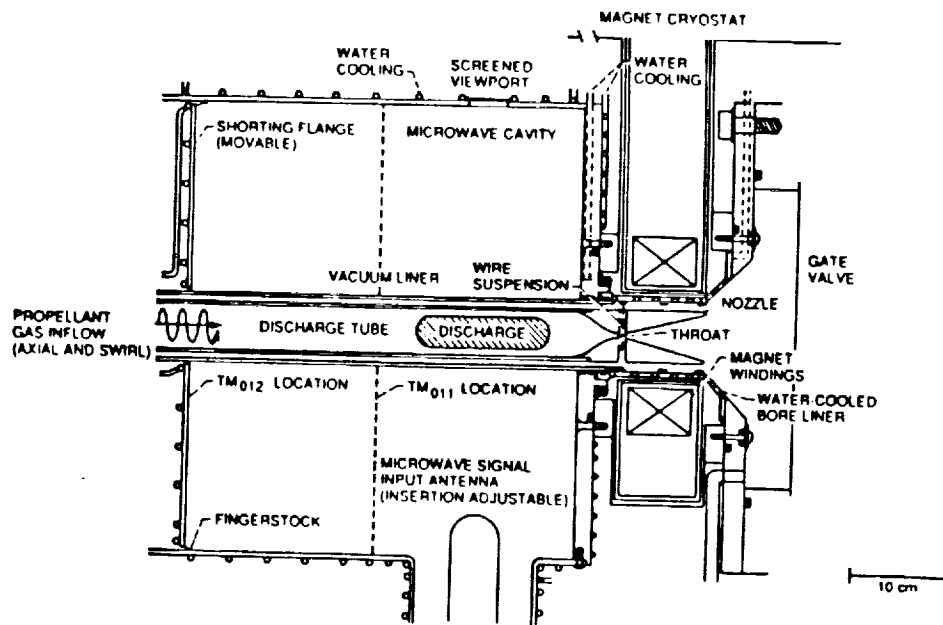


Figure 15. Schematic of the microwave electrothermal thruster (MET) concept showing the microwave cavity and the magnetic nozzle.

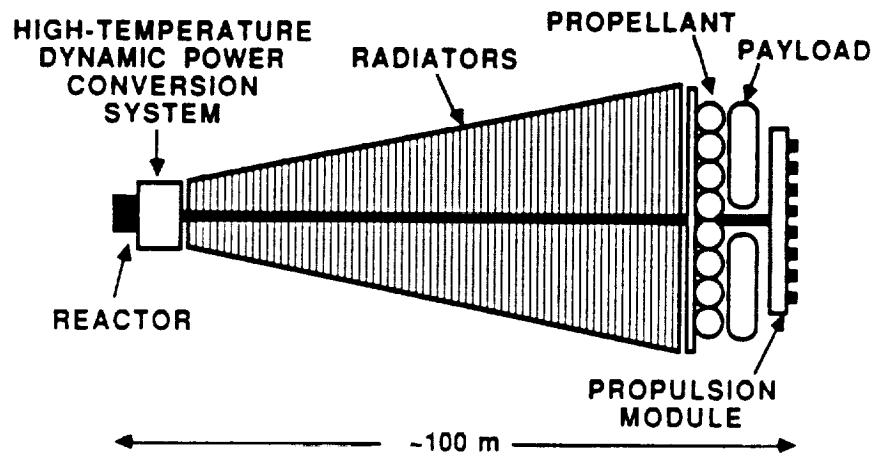


Figure 16. Conceptual diagram for a 100-MWe-class nuclear electric propulsion (NEP) system.

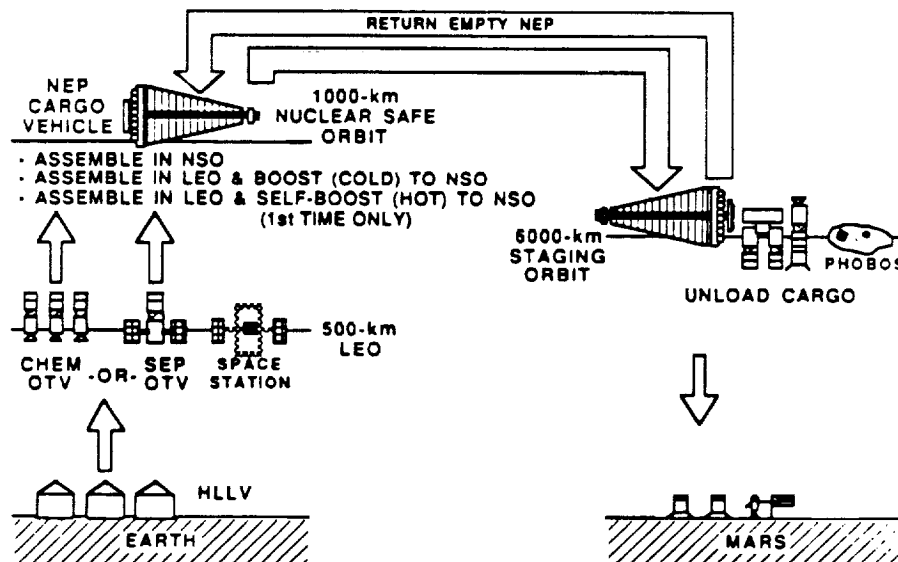


Figure 17. Mission scenario assumed for the study of the 100-MWe-class NEP system.

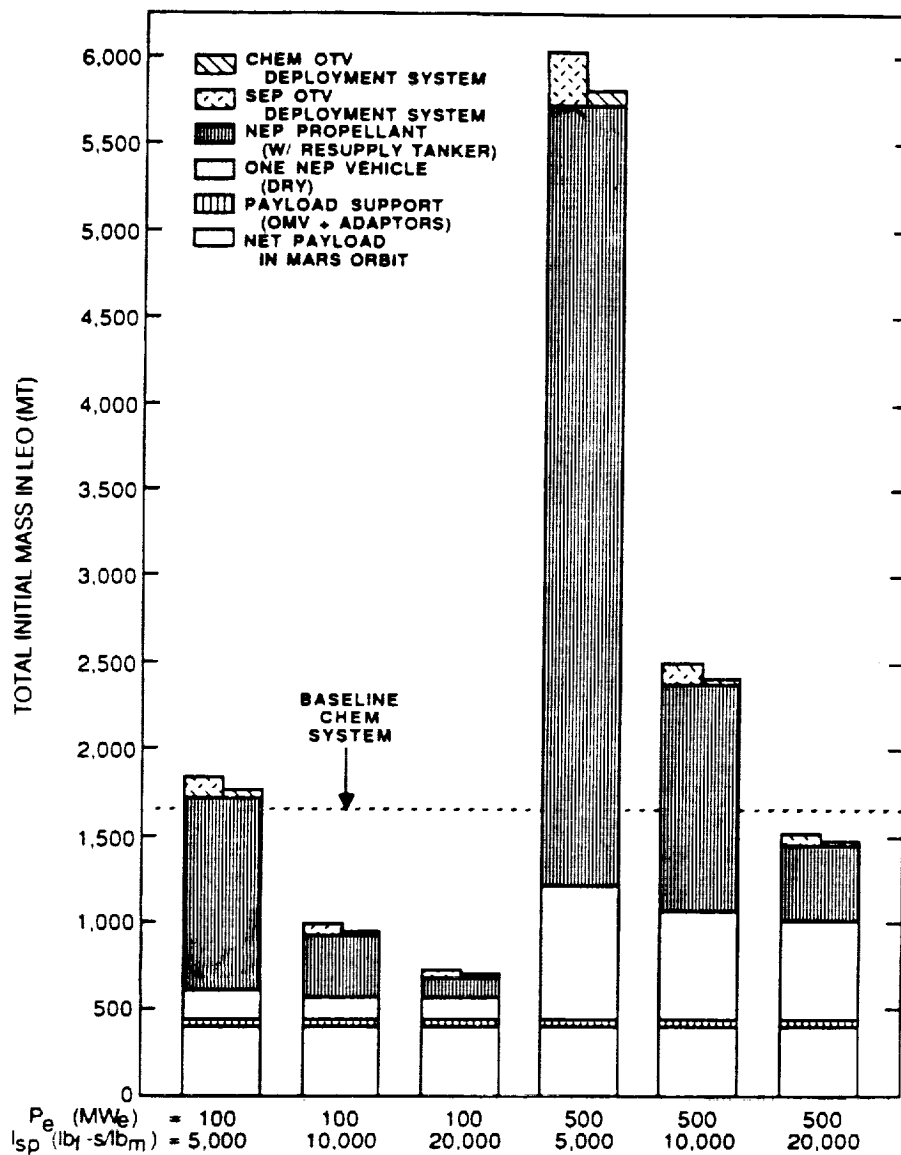


Figure 18. Mass breakdown for the NEP system as a function of power level and specific impulse. The baseline chemical system is shown for comparison.

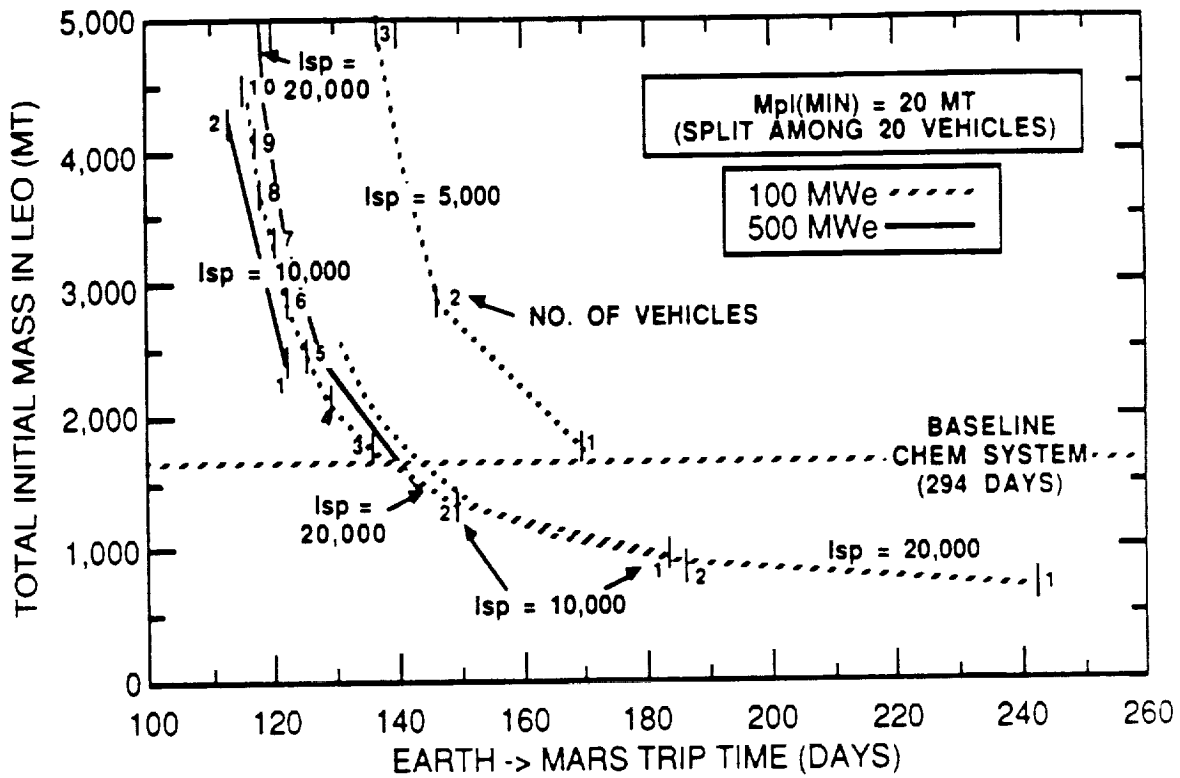


Figure 19. Initial mass into low Earth orbit as a function of the Earth-to-Mars trip time for a 100-MWe-class nuclear electric propulsion (NEP) system.



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16. Abstract The NASA Office of Aeronautics, Exploration and Technology (OAET) supports a research and technology (R&T) program in electric propulsion to provide the basis for increased performance and life of electric thruster systems which can have a major impact on space system performance, including orbital transfer, stationkeeping, and planetary exploration. The program is oriented toward providing high-performance options that will be applicable to a broad range of near-term and far-term missions and vehicles. The program, which is being conducted through the Jet Propulsion Laboratory (JPL) and Lewis Research Center (LeRC) includes research on resistojets, arcjets, ion engines, magnetoplasmadynamic (MPD) thrusters, and electrodeless thrusters. Planning is also under way for nuclear electric propulsion (NEP) as part of the Space Exploration Initiative (SEI).					
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